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CERAMIC ROTOR BLADE DEVELOPMENT

PART I - CERAMIC THERMAL BARRIER COATINGS

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Fourth Semi-Annual Technical Report
Research Project 421-1
Contract Manager: Dr. Art Cohn

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EXECUTIVE SUMMARY

The objective of the "Ceramic Rotor Blade Development Program" as currently revised is the design and fabrication of blade root attachments from brittle refractory candidate materials for evaluation through spin tests as a proof of design attachment concepts and material capability, leading to the full development of ceramic rotor blades and cantilevered vanes intended for industrial gas turbines. At the request of EPRI, the program objective was modified further in 1976 to include ceramic thermal barrier coatings as a possible candidate for raising operating temperatures of metal vanes and blades. It should be emphasized that the end result of the present work will not necessarily represent the design of component parts specifically for turbogenerator machinery, but rather will lead to an established design philosophy for turbine blades and cantilevered vanes manufactured from ceramics.

The present effort has been subdivided into three principal tasks. These include:

Task I - Rotor Blade Development

Task II - Thermal Barrier Coatings

Task III - Cantilevered Vane Development

This report (Part I) provides progress made on Task II - Thermal Barrier Coatings. A subsequent report (Part II) will be issued in July after completion of Task I.

Task II - Thermal Barrier Coatings

The potential benefits from the use of thermal barrier coatings in industrial gas turbines are reduction of metal component temperatures by 200°F - 400°F or reduction in cooling air requirements for increased turbine efficiency. This report summarizes 1) the behavior of duplex thermal barrier coating systems, consisting of a MCrAlY (M = Ni or Ni/Co) bond

coat and a Y_2O_3-stabilized ZrO_2 overcoat, on Udimet-500 in simulated gas turbine environments at 2000°F gas temperature, and 2) the mechanical properties of the coating system from R.T. to 1500°F.

The hot corrosion/erosion testing in pressurized test passages included coating systems prepared by NASA-Lewis Research Center and Union Carbide Corporation, Linde Division. Under the environmental conditions employed, which included cooled and uncooled specimen configurations, it has been found that the coating systems undergo external cracking, chipping and spalling of the oxide overcoat. The cause of the problem has not yet been clearly defined, but observations suggest the following interactive factors:

1) Destabilization of the ceramic coating in fuel containing Na, V, S, and P. The as-deposited stabilized cubic ZrO_2-Y_2O_3 phase → monoclinic ZrO_2, which is a detrimental phase. Monoclinic ZrO_2 undergoes phase transformation during heating and cooling with an associated disruptive volume change.

2) Formation of secondary phases due to reactions between the combustion products and the ceramic coating constituents; e.g., Y_2O_3 reacts with phosphorous to form yttrium phosphate.

3) Mismatch in thermal expansion between the ceramic and bond coat.

4) Low strength of the ceramic coating.

Mechanical property testing was curtailed in February 1977 to concentrate on the hot corrosion work. Preliminary data was obtained for tensile strength, creep rupture life and high cycle fatigue. All properties, except for the A ratio = ∞ in high cycle fatigue tests, either met or exceeded the properties of the uncoated base metal.

The result obtained to date clearly indicates the need for further development work and a better understanding of thermal barrier coating behavior in high temperature corrosive environments before these coating systems can be employed for industrial turbine application.

TABLE OF CONTENTS

INTRODUCTION

The objective of the "Ceramic Rotor Blade Development Program" as currently revised is the design and fabrication of blade root attachments from brittle refractory candidate materials for evaluation through spin tests as a proof of design attachment concepts and material capability, leading to the full development of ceramic rotor blades and cantilevered vanes intended for industrial gas turbines. At the request of EPRI, the program objective was modified further in 1976 to include ceramic thermal barrier coatings as a possible candidate for raising operating temperatures of metal vanes and blades. It should be emphasized that the end result of the present work will not necessarily represent the design of component parts specifically for turbogenerator machinery, but rather will lead to design techniques for turbine blades and cantilevered vanes manufactured from ceramics.

The present effort has been subdivided into three principal tasks. These include:

Task I - Rotor Blade Development
Task II - Thermal Barrier Coatings
Task III - Cantilevered Vane Development

In Task I the plan is to complete spin testing of root forms at R.T., complete the failure analysis and stress analysis work needed to correlate analytical results with spin test results, and together with a 2-D parametric study, determine the likely design changes required for full-scale rotor blade development.

In Task II the plan is to complete corrosion testing of cooled and uncooled thermal barrier specimens obtained from NASA Lewis and Linde Company and complete microanalysis of tested specimens in order to determine possible corrosion problems with these structures; and conduct selected mechanical property tests to determine effects of thermal barrier coatings on the base alloy properties.

In Task III the plan is to initiate design and analysis studies of second-stage cantilevered ceramic vane attachment.

With the verbal approval of EPRI (Dr. Art Cohn), this Fourth Semi-Annual Technical Report will be presented in two parts as follows:

PART I - Task II, Ceramic Thermal Barrier Coatings, issued May 15, 1977.

PART II - Task I, Ceramic Rotor Blade Development, to be issued in July after completion of the task.

TASK II - CERAMIC THERMAL BARRIER COATINGS

1.1 PERFORMANCE OF THERMAL BARRIER COATINGS IN SIMULATED GAS TURBINE ENVIRONMENTS (S. C. Singhal and S. Y. Lee)

I. INTRODUCTION

There is considerable incentive to increase gas inlet temperature in stationary gas turbines to improve thermal efficiency and increase power output for a given engine size. Because of the difficulty in achieving major improvements in temperature capabilities of metallic alloy systems and the significant losses in turbine efficiency in using airfoil cooling, serious consideration is being given to coating the metal components with a thermally insulating ceramic material for maintaining metal components at much lower temperatures than the hot combustion gases. The potential benefits from the use of such thermal barrier coatings are quite large and include reduction of metal temperatures by about 200-400°F, or reduction in cooling air requirements for increased turbine efficiency. This report summarizes the behavior of duplex thermal barrier coating systems, consisting of a MCrAlY (M = Ni or Ni/Co) bond coat and the Y_2O_3-stabilized ZrO_2 overcoat, on Udimet-500 in simulated gas turbine environments at 2000°F gas temperature.

II. EXPERIMENTAL

1. The Experimental Equipment

The behavior of duplex thermal barrier coating systems has been studied in the pressurized turbine test passage which simulates the operational environment of a stationary gas turbine. Fig. 1 shows a schematic diagram of this test facility. In this facility, air compressed to about 100 psi is preheated in an indirectly fired air heater to ∿600°F and fed into a Hastelloy-X combustor. Fuel is injected into the primary combustion zone through a bayonet-mounted fuel nozzle in the dome-shaped front portion of the combustor and burned with the compressed air. Combustion gases are then mixed with secondary cooling air, the amount of which is adjusted to attain the desired gas temperature. At the combustor exit, an array of 16 thermocouples measures the temperature of the combustion gases.

Downstream of the thermocouple array, a transition section reduces the flow area and increases the gas velocity to ∿500 fps. At this location, the coated specimens are exposed to the high velocity hot combustion gases. A test section holds a set of eight 1/4" diameter uncooled solid cylindrical pins, or two air-cooled cylindrical sleeves 1" in diameter and 2" long (Fig. 2). These cylindrical sleeves have 12 holes drilled lengthwise to various depths to accommodate thermocouples which measure the metal surface temperature. At the other end of the test section, a water-cooled barrier plate acts as a damper valve and maintains a pressure of 3 atm in the passage. The exhaust gases are cooled by a water spray and vented to the atmosphere through an extensive muffler system to reduce noise to acceptable levels.

The basic fuel used for all the experiments described in this report was No. GT-2 diesel oil which satisfied the following specifications:

sulfur	0.35 wt% (maximum)
ash	0.01 wt% (maximum)
water + sediment	nil
carbon residue	0.2 wt% (maximum)

The sulfur level in this fuel was increased to 0.5 wt% by adding appropriate quantities of a sulfur compound. Other contaminants, e.g., Na and V, were added to the desired levels.

2. The Coating Systems

The duplex coatings consisting of a MCrAlY (M = Ni or Ni/Co) bond coat and the Y_2O_3-stabilized ZrO_2 overcoat on Udimet-500 specimens were obtained from two sources:

(a) NASA-Lewis Research Center

(b) Union Carbide Corp. - Linde Division.

Both the bond coat and the oxide overcoat in each of the coating systems were deposited by plasma spraying. The Linde coating was deposited using an inert gas shroud surrounding the plasma spray effluent in order to eliminate any oxidation of the bond coat alloy powder as it was sprayed. No such protective shield was used by NASA in depositing the coatings.

The bond coat in the NASA coating had a nominal composition of 16 wt% Cr, 5 wt% Al, 0.6 wt% Y and the balance Ni, while the overcoat was zirconia stabilized with 12 wt% Y_2O_3. Metallographic cross-sections of a typical NASA coating are shown in Figs. 3 and 4. The bond coat in

these coatings was found to vary from ∿0.5 to 4 mil in thickness around the circumference of the specimen, and at a few points, there was practically no bond coat present. Electron microprobe analysis revealed the presence of large inclusions of alumina, which appear as dark regions in Fig. 4, in the bond coat. The stabilized ZrO_2 overcoat was relatively uniform in thickness, ∿14 mil. This oxide coating was found to be mainly the cubic ZrO_2-Y_2O_3 phase. However, it did contain a minor second phase rich in yttrium with traces of silicon.

The bond coat in the Linde coating consisted of a cobalt-base alloy with 32 wt% Ni, 21 wt% Cr, 7.5 wt% Al and 0.5 wt% Y, while the overcoat was again zirconia stabilized with 12 wt% Y_2O_3. The metallographic cross-section of a typical Linde coating is shown in Fig. 5. Both the bond coat and the oxide coating in these Linde coatings were found to be fairly uniform in thickness; the bond coat being ∿6.5 mil and the oxide coating ∿12 mil in thickness. The surface roughness of the Linde coating averaged 240 micro-inch, rms, as compared to 400 micro-inch for the NASA coating.

III. RESULTS

1. Uncooled Solid Test Specimens

The experimental conditions under which uncooled coated specimens have been tested are summarized in Table I, after which the results from each test are described.

TABLE I

Gas Temperature	Fuel Additives	Coating Type
(a) 1650°F	5 ppm Pb	NASA
(b) 1650°F	3 ppm Na, 5 ppm V	NASA, Linde
(c) 1800°F	10 ppm Na, 18 ppm Cl, 1.5 ppm Mg	NASA, Linde

(a) The coating on the NASA-coated Udimet-500 specimen exposed to hot combustion gases obtained by burning GT-2 diesel with 5 ppm Pb remained intact and free of any macrocracking or spalling even after 101 hrs of exposure. The surface of the coated specimen after 101 hrs of exposure is shown in Fig. 6 along with an as-coated untested specimen. The coating surface picked up small amounts of deposits from the combustion gases during exposure and showed a net weight gain of 11.6 mg in 101 hrs. These deposits were identified by x-ray diffraction analysis to be predominantly barium sulfate. Barium was a constituent of the fuel used, which is added by some fuel manufacturers to prevent gum formation during fuel storage and to reduce excessive smoke formation on combustion. Linde coating was not tested under these conditions because of its unavailability at the time of the test.

(b) One each NASA- and Linde-coated specimen was exposed to hot combustion gases at 1650°F using GT-2 diesel oil with 3 ppm Na and 5 ppm V. The surface appearances of these exposed specimens (4 views of each specimen) are illustrated in Figs. 7 and 8, which show that the oxide coating on both the NASA- and the Linde-coated specimens severely chipped and spalled off very early during the exposure. The oxide coating on the NASA-coated specimen was almost completely gone at 62 hrs of exposure,

while that on the Linde-coated specimen lasted only 31 hrs of exposure. The oxide coating appeared to have separated from the MCrAlY bond coat in both cases; however, no oxidation or corrosion of the MCrAlY bond coat was apparent in the preliminary analysis of the exposed specimens. The oxide coating thus appears to crack and subsequently spall off due to thermal expansion mismatch between the different layers.

(c) One each NASA- and Linde-coated specimen was exposed at 1800°F to hot combustion gases obtained by burning GT-2 diesel fuel with 10 ppm Na, 18 ppm Cl and 1.5 ppm Mg. The oxide coating on the NASA-coated specimen was found to be free of cracks and spalling after 78 hrs of exposure; but with further exposure, it severely cracked and appeared to have completely separated from the specimen as shown in Fig. 9. A metallographic cross-section of the exposed specimen after this 78 hrs of exposure is shown in Fig. 10, which reveals that the oxide coating separated from the NiCrAly bond coat. However, no corrosion of this bond coat was apparent.

The oxide coating on the Linde-coated specimen also chipped and spalled off in small pieces and was almost completely removed from some regions of the specimen in 87 hrs of exposure to the hot combustion gases. The surface appearance of the Linde-coated specimen (4 views of the specimen) after 87 hrs of exposure is shown in Fig. 11. The metallographic cross-sections from two different regions of the exposed specimen, shown in Fig. 12, revealed that the oxide coating in this case did not separate from the bond coat but rather cracked and chipped off within its own thickness.

2. Cooled Test Specimens

Cooled tests were run on both NASA- and Linde-coated specimens using GT-2 diesel fuel with 5 ppm Na and 2 ppm V at a gas temperature of 2000°F. Two tests have been completed so far as described below:

(a) In the first test, one NASA-coated specimen and one uncoated Udimet-500 specimen were exposed to the hot combustion gases at 2000°F. Both specimens were air-cooled to keep the average metal temperature at ∿1550°F. The maximum and minimum temperatures experienced by the different regions of the metal surface of the NASA-coated specimen during the course of the test are shown in Fig. 13. This temperature profile shows that some regions of the metal surface experienced temperatures as high as 1805°F during the course of the test. The NASA-coated specimen started exhibiting some spalling of the oxide coating as early as 21 hrs of exposure to combustion gases, and the amount of spalling increased with increasing exposure time. The front surface of the specimen after 46.5 hrs of exposure is shown in Fig. 14. The amount of oxide spalling was observed to be greatest in the bottom front of the specimen, which is the region of the highest metal surface temperature. The metallographic cross-sections from four different regions around the circumference of the specimen are shown in Figs. 15 and 16. At about 1/2" from the bottom of the specimen (Location A in Fig. 14), the metallographic cross sections (Fig. 15) show that the oxide coating separated from the NiCrAlY bond coat both at the front and the right side of the specimen. However, the coating did remain intact on the backside of the specimen. At the bottom of the specimen (Location B in Fig. 14), however, not only the oxide coating spalled off completely

throughout the circumference of the specimen but some NiCrAlY bond coat also spalled off (Fig. 16). As a result of this removal of NiCrAlY bond coat, severe corrosion of the Udimet-500 substrate has occurred.

The surface of the exposed specimen was examined by x-ray diffraction analysis, which revealed that some transformation of the originally cubic ZrO_2 phase to monoclinic phase in the oxide coating has occurred during exposure to the hot combustion gases. Furthermore, the x-ray diffraction analysis indicated the strong possibility of the presence of appreciable amounts of yttrium phosphate in the surface oxide coating. This suggests that some yttrium in the original cubic phase reacts with phosphorous* in the combustion gases to form yttrium phosphate with simultaneous formation of the Y_2O_3-depleted monoclinic phase. Such reactions and phase transformations will result in volumetric changes in the oxide coating which could also cause spalling of the oxide in addition to thermal expansion mismatch and thermal gradients in the oxide coating. Corrosion itself is not believed to be a significant factor in the degradation of the oxide thermal barrier coating for the short exposure times of this particular test run. However, significant amounts of Na_2SO_4 were found to be present in the oxide coating after exposure to the hot combustion gases. This suggests that the bond coat and the substrate could experience hot corrosion after extended periods of exposure to the hot combustion gases.

*Phosphorous has been found to be present in GT-2 diesel oil at a level of few ppm. This gets converted to P_2O_5 gas in the combustion gases.

(b) The second cooled test, on both NASA- and Linde-coated specimens, was also run at 2000°F gas temperature using the same fuel and additives, but the amount of cooling air was increased to keep the metal surface temperature to a maximum of 1650°F. This resulted in an average metal surface temperature of ∿1450°F. The maximum and minimum temperatures experienced by the different regions of the metal surface of the NASA- and the Linde-coated specimens are shown in Fig. 17.

After 37 hours of exposure, the oxide coating on the NASA-coated specimen was found intact while that on the Linde-coated specimen experienced some spalling as shown in Fig. 18. However, with further exposure to a total of 63.5 hrs, the NASA coating also showed oxide spalling while the amount of oxide spalling increased on the Linde-coated specimen. The severe spalling on the Linde-coated specimen is illustrated in Fig. 19 by four views of the specimen after 63.5 hrs of exposure. The metallographic cross-sections from four different locations around the circumference of this exposed specimen are shown in Fig. 20. These reveal that numerous cracks develop in the oxide coating during exposure to the hot combustion gases which, in turn, cause spalling of the oxide coating. Again, no corrosion of the bond coat is apparent in these micrographs.

As shown in Fig. 21, similar severe spalling of the oxide coating on the NASA coated specimen has been found to occur with further exposure to a total of 100 hrs.

In order to separate the effects of the temperature and the corrosive contaminants on oxide spalling, two more tests are being run on NASA- and Linde-coated air-cooled specimens under the following conditions:

(i) At 2000°F gas temperature keeping the metal surface temperature to a maximum of 1650°F, using clean GT-2 diesel oil with no additives.

(ii) At 2000°F gas temperature keeping the metal surface temperature to a maximum of 1450°F, using GT-2 diesel oil with 5 ppm Na and 2 ppm V.

1.2 MECHANICAL TESTING (W. Hays S. T. Scheirer, D. M. Moon)

In mid February the planned work on mechanical testing of ceramic thermal barrier specimens was curtailed at the request of the EPRI Program Manager, Dr. A. Cohn. All mechanical test efforts were terminated on February 21, 1977 at GSD except two creep rutpure tests and one high cycle fatigue test which were in progress on that date. Specimen coating work in progress at that time with Linde Company was continued through completion.

The sample disposition as of February 22, was as follows:

Test	NASA Specimens	Linde Specimens
1. Tensile	4 Tests completed	4 Specimens not received
2. Creep rupture	2 Tests completed 2 Tests running 3 Tests cancelled 3 Specimens extra	10 Specimens not received
3. High Cycle Fatigue	5 Tests completed 1 Test running 6 Tests cancelled	12 Specimens not received
4. Thermal Fatigue*	4 Non-coated specimens @ R&D for calibration and control 2 Coated specimens @ R&D	4 Non-coated specimens @ R&D for calibration and control 2 Coated Specimens not received
5. Low Cycle Fatigue	6 Tests Cancelled	6 Specimens not received

The Linde specimens not yet received have been shipped and should arrive in the near future.

*No thermal fatigue tests are running or completed.

The following results were obtained during mechanical testing of NASA coated thermal barrier specimens.

Tensile Tests

Base Mat'l	Temp. °F	Uncoated (Design Manual) UTS ksi	.2YS ksi	(Actual) UTS ksi	.2YS ksi
U-500	78	126	92	140	114
U-500	1400	123	92	132	97
X45	78	102	65	105	63
X45	1400	56	27	74	30

Creep Rupture*

Base Mat'l	Temp. °F	Stress ksi	Life (Design Manual) hrs.	Life (Actual) hrs.
U-500	1475	50	200	268
X45	1500	25	39	70

High Cycle Fatigue

Base Mat'l	Temp. °F	Stress ksi	A Ratio	Life (Expected) Cycles	Life (Actual) Cycles
U-500	1000	$\pm$ 35	∞	2.4×10^6	$.701 \times 10^6$
U-500	1000	$\pm$ 32	∞	10.0×10^6	$.905 \times 10^6$
U-500	1000	$\pm$ 30	∞	100.0×10^6	12.784×10^6
U-500	1000	80 $\pm$ 20	.25	3.5×10^5	161.24×10^5
U-500	1000	84 $\pm$ 24	.25	9.0×10^4	65.1×10^4

*Elongation and reduction of area meets specification minimums.

As illustrated above, all properties except the A ratio = ∞, High Cycle Fatigue Results, either met or exceeded the properties of the uncoated base material. It should be noted, however, that stresses were based on the uncoated specimen dimensions.

With respect to the A ratio = ∞, High Cycle Fatigue Results, some questions remain concerning the reasons for the lower than expected properties. Two effects could have contributed to this condition:

1. Coating cracked during cycling with subsequent propagation into the base metal (this could be confirmed by metallography).
2. Coating thickness variation caused poor alignment in grips causing locally high stresses in this axial test and premature failure (hard to prove but very possible).

Thermal Shock Resistance of Thermal Barriers

During the operation of cooled gas turbine hardware containing thermal barriers several factors may lead to significant stresses being generated in the ceramic coating. During transient operation such as hot starts or trip outs, rapid changes in gas path temperatures will impose thermal gradients within the coating due to the more rapid response of the coating surface. During steady-state operation, stresses will also be imposed on the barrier due to the steep temperature gradients existing across it. Differential thermal expansion effects in the three-layer composite of ceramic, bond coat and base metal could also be a significant source of stress during both steady state and transient operation. Of these possible sources of stress, it is considered that the most critical will be the tensile stresses developed during a trip out with the compressor at full power.

In order to evaluate the capability of the ceramic barrier to withstand these transient tensile stresses a test program has been developed based on the works of Manson and Smith.* The test involves determining the maximum temperature difference that a specimen can withstand without cracking after quenching into a bath of known heat transfer characteristics. From this information the maximum tensile stress generated in the barrier can be calculated and used to define the critical stress required for cracking. Once this critical stress has been established then it represents the fracture strength of the material and an evaluation can be made of any desired source of stress, using assigned heat transfer coefficients and thermal transients.

To date the specimen for testing has been designed, and eight specimens have been machined from U-500 and X-45 alloys. Machined specimens have been sent to Linde and NASA for application of their respective thermal barrier systems and the NASA-coated specimens have been received back.

In order to determine the heat transfer coefficient of the quenching medium as well as the stresses developed during the test a two-dimensional isoparametric finite element computer program was developed for the specimen geometry being used. Using the program, the transient temperature analysis required to determine the heat transfer coefficient of the quenching medium was completed.

1.3 PROCESS DEVELOPMENT (R. Harry)

Meetings were conducted with Linde and TRW to become knowledgeable of their coating processes and to acquaint them with the potential application of thermal barrier coatings to gas turbine hardware. This work was curtailed in February 1977.

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* S.S. Manson and R. W. Smith, "Quantitative Evaluation of Thermal Shock Resistance," Trans ASME (1956), 533.

ACKNOWLEDGEMENT

The authors acknowledge Dr. Stanley Levine, NASA Research Center for his cooperation in preparing thermal barrier coatings and helpful discussion.

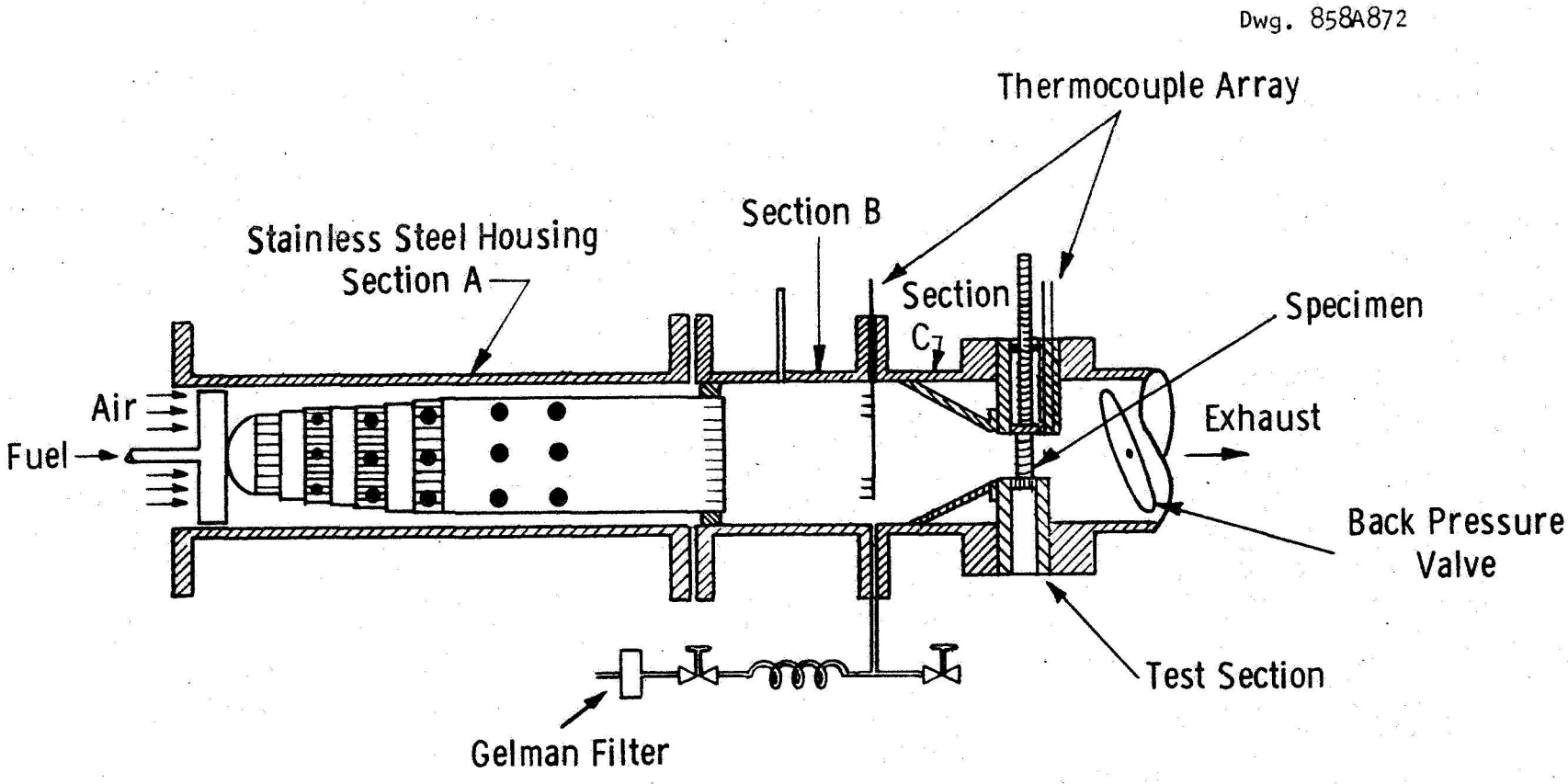


Fig. 1—Turbine test passage before modification

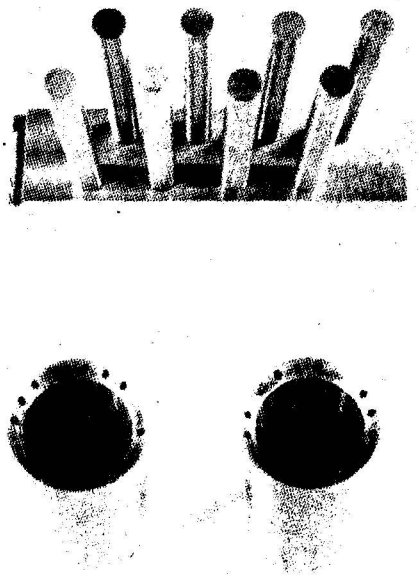


Fig. 2 - 1/4" diameter solid pins for uncooled tests (top);
1" diameter cylindrical sleeves for cooled tests (bottom).

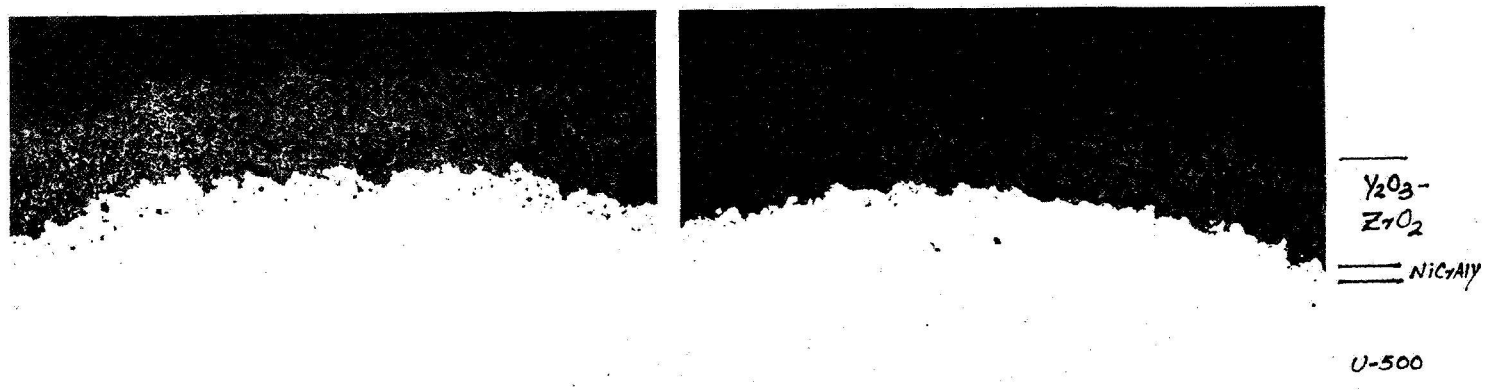


Fig. 3 - Metallographic cross-sections of the NASA coating showing non-uniformity of the NiCrAlY bond coat (50X).

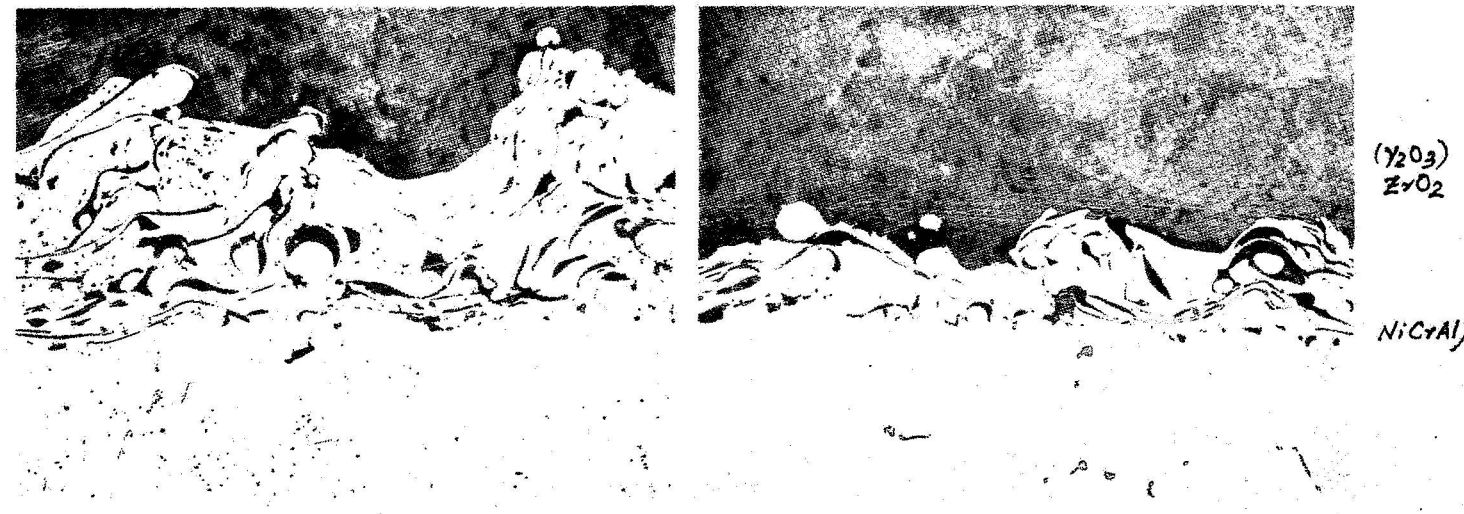


Fig. 4 - Metallographic cross-sections of the NASA coating showing details of the NiCrAlY bond coat (500X).

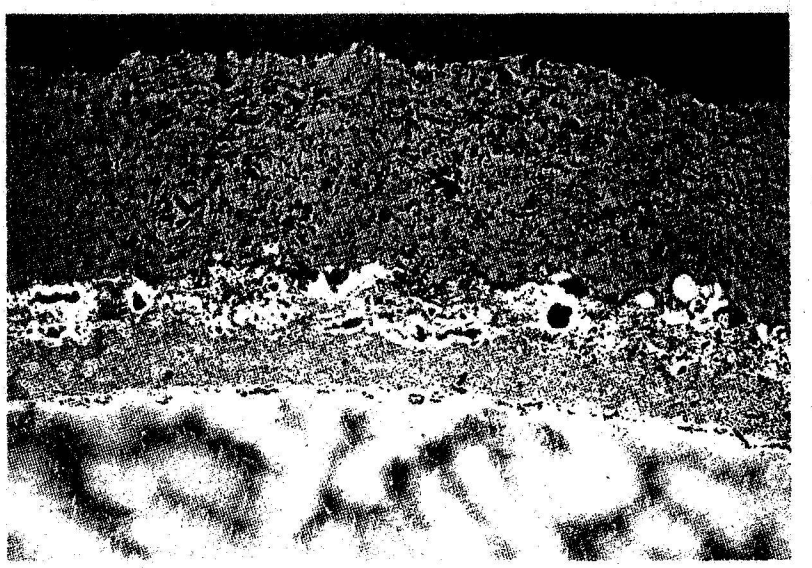


Fig. 5 - Metallographic cross-section of the Linde coating showing the bond coat and the oxide thermal barrier coating (100X).

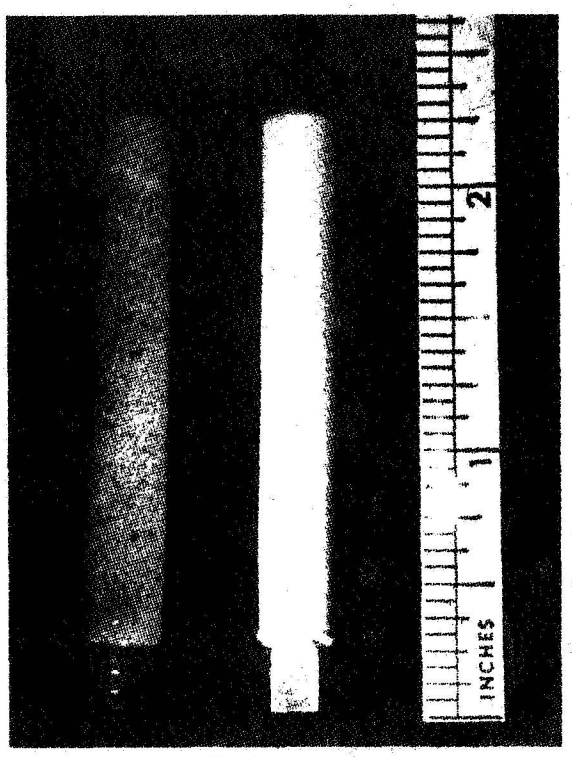


Fig. 6 - The NASA-coated specimens before (right) and after (left) 101 hours of exposure at 1650°F with 5 ppm Pb in the fuel.

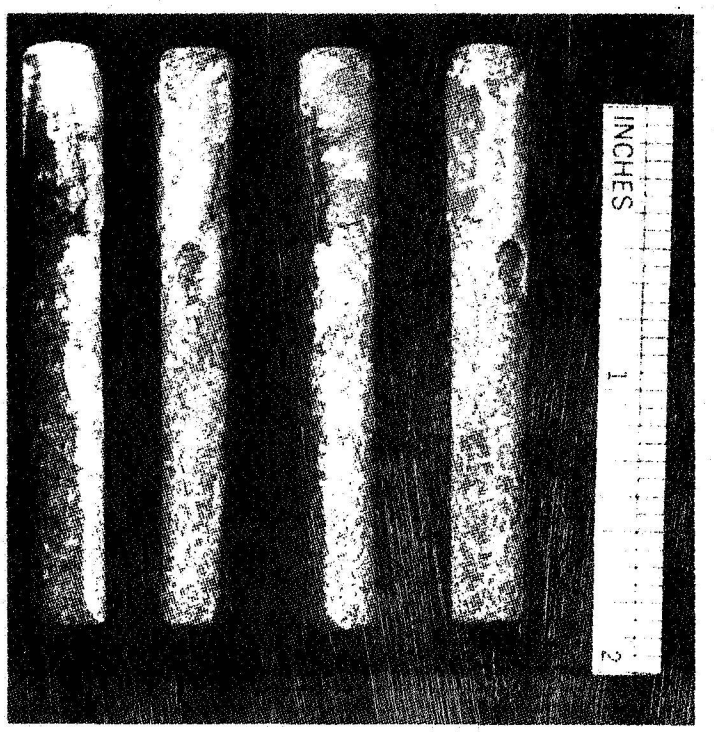


Fig. 7 - The NASA-coated specimen after 62 hours of exposure at 1650°F with 3 ppm Na and 5 ppm V in the fuel.

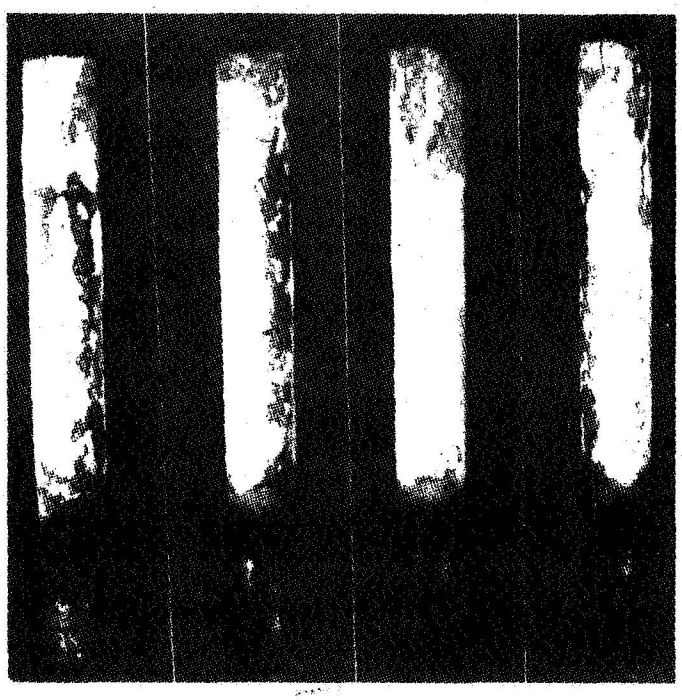


Fig. 8 - The Linde-coated specimen after 31 hours of exposure at 1650°F with 3 ppm Na and 5 ppm V in the fuel.

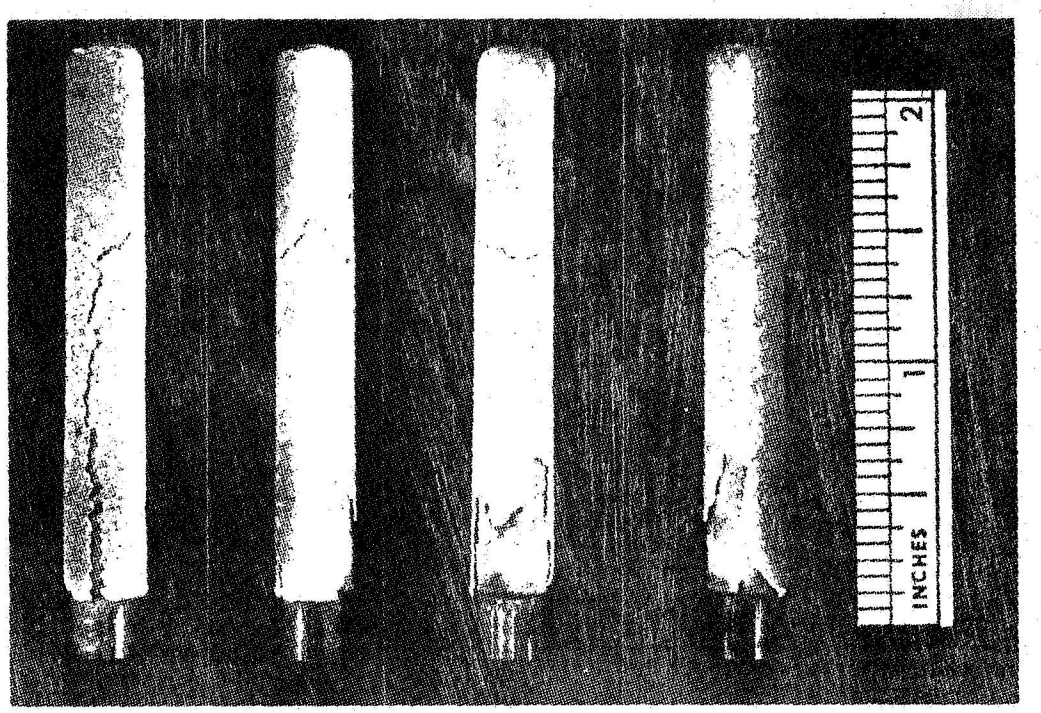


Fig. 9 - The NASA-coated specimen after 78 hours of exposure at 1800°F with 10 ppm Na, 18 ppm Cl, and 1.5 ppm Mg in the fuel.

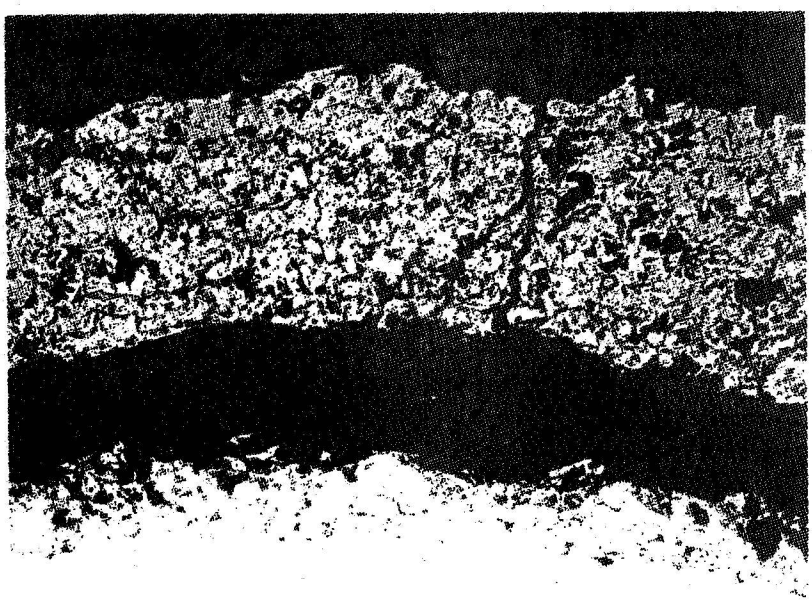


Fig. 10 - Metallographic cross-section of the NASA-coated specimen after 78 hours of exposure at 1800°F with 10 ppm Na, 18 ppm Cl, and 1.5 ppm Mg in the fuel.

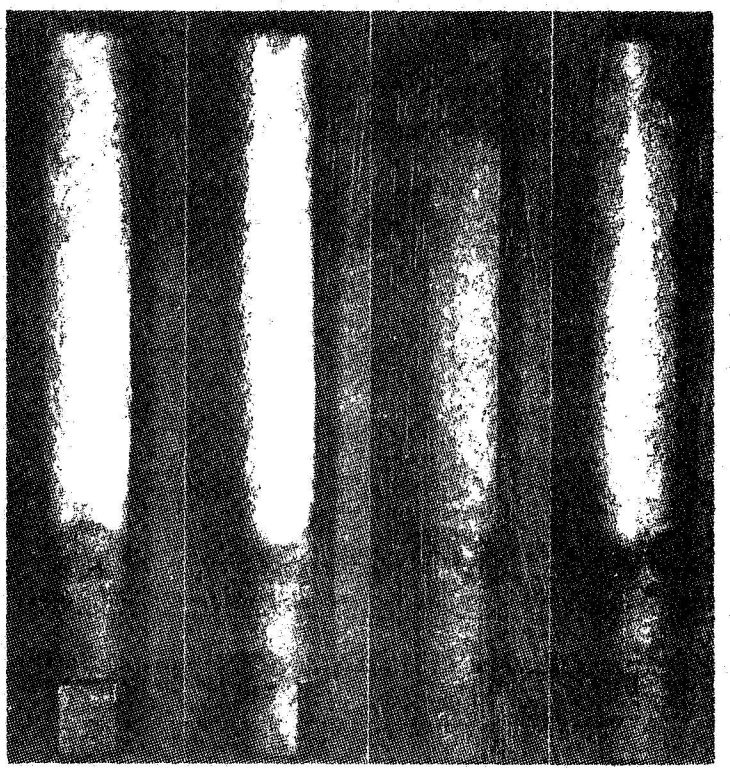


Fig. 11 - The Linde-coated specimen after 87 hours of exposure at 1800°F with 10 ppm Na, 18 ppm Cl, and 1.5 ppm Mg in the fuel.

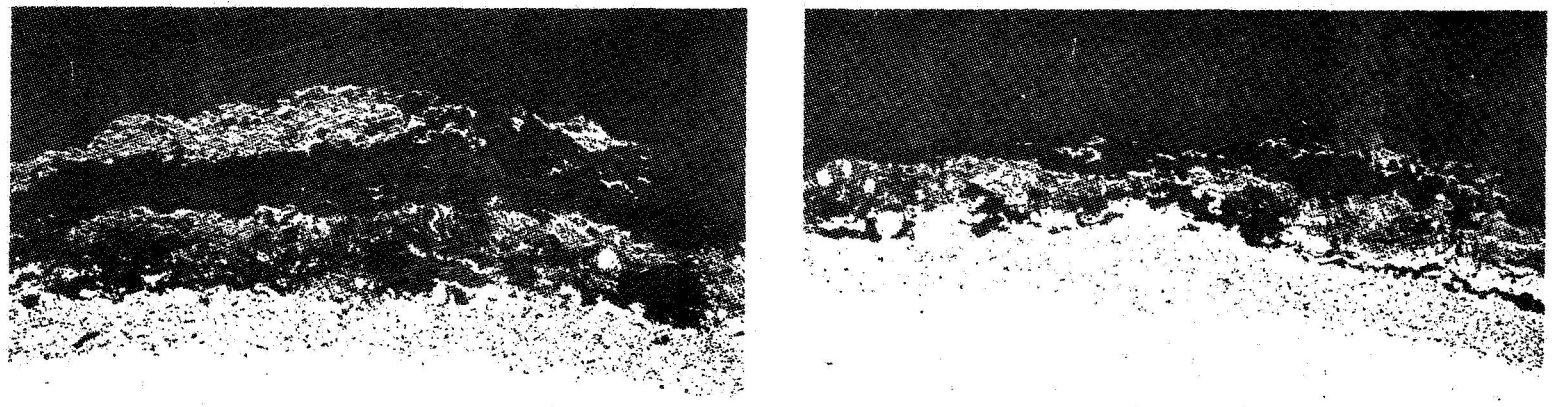


Fig. 12 - Metallographic cross-sections from two regions of the Linde-coated specimen after 87 hours of exposure at 1800°F with 10 ppm Na, 18 ppm Cl, and 1.5 ppm Mg in the fuel.

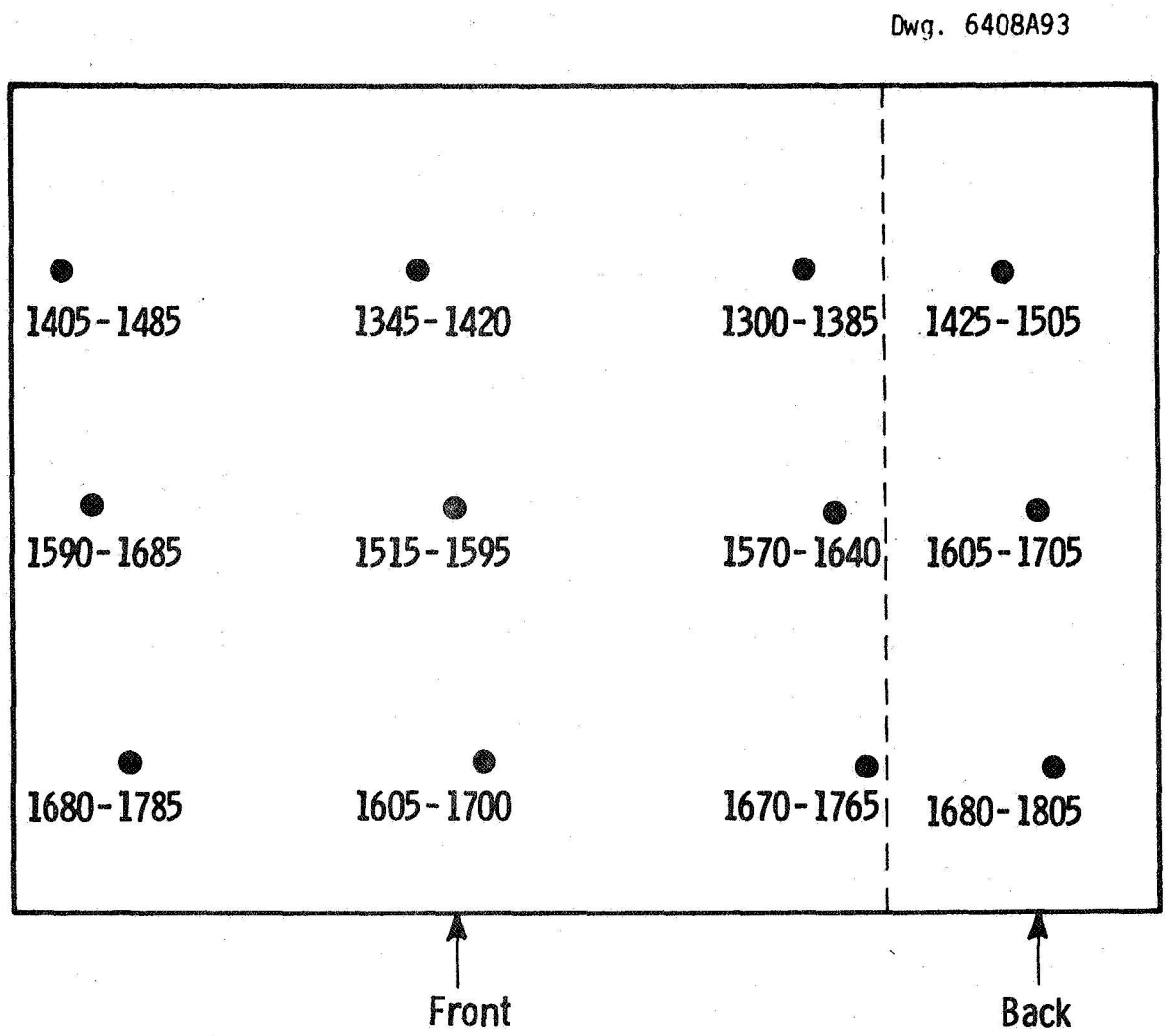


Fig. 13 — Temperatures (in °F) at the metal surface of the NASA-coated air-cooled specimen in the pressurized turbine test passage at 2000°F gas temperature

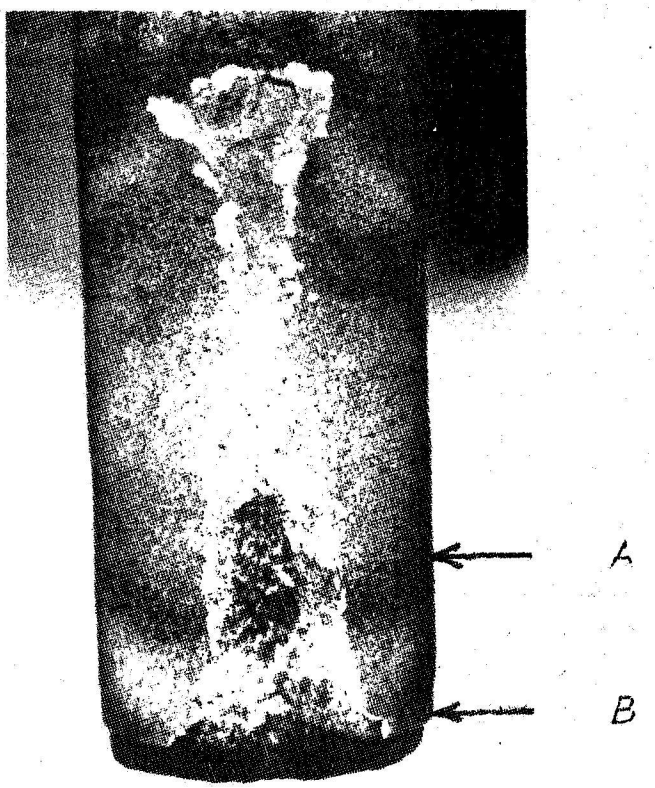


Fig. 14 - The front view of the NASA-coated specimen after 46.5 hours of exposure with average metal surface temperature of 1550°F.

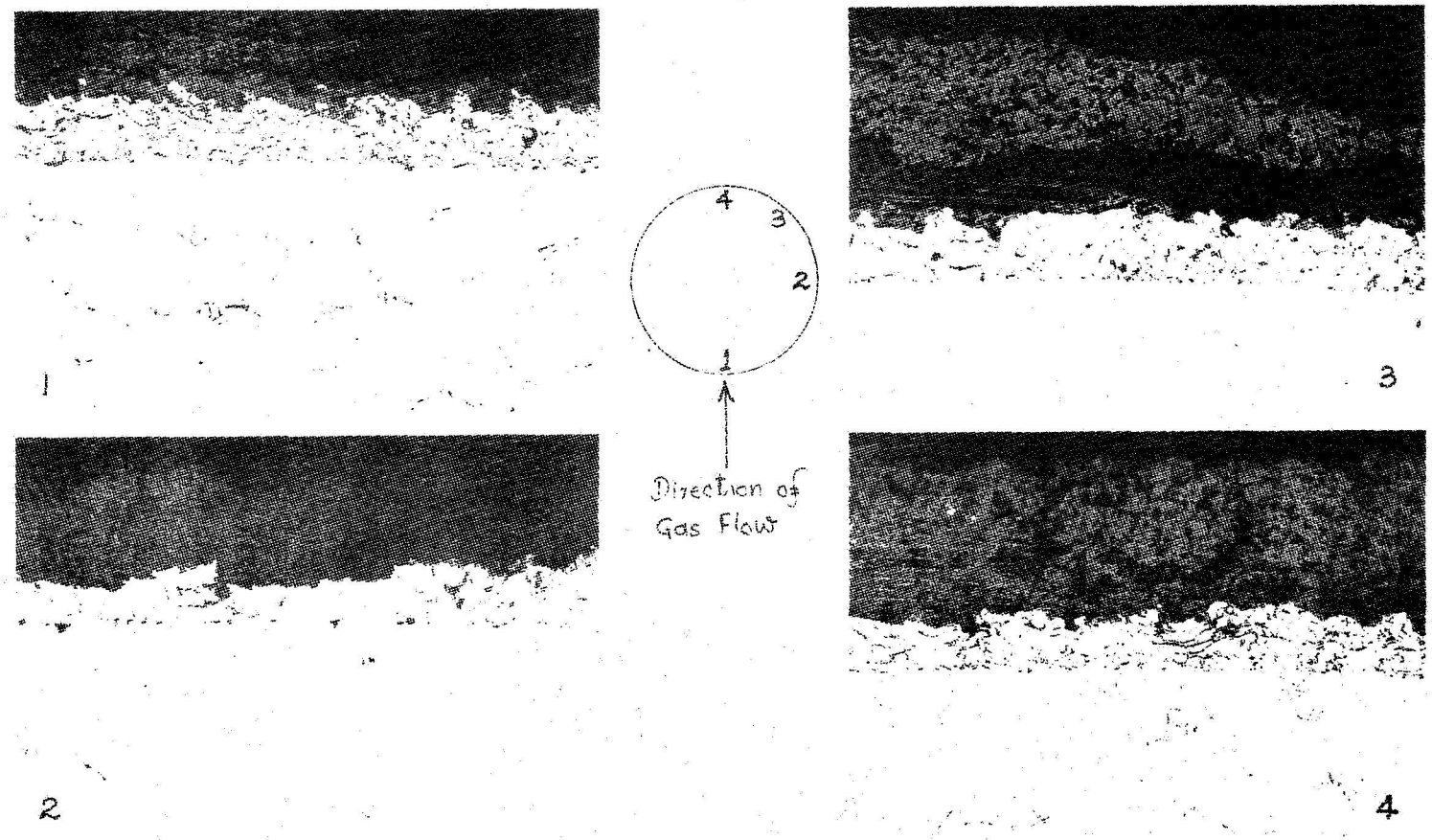


Fig. 15 - Metallographic cross-sections of the NASA-coated specimen after 46.5 hours of exposure; location A (100X).

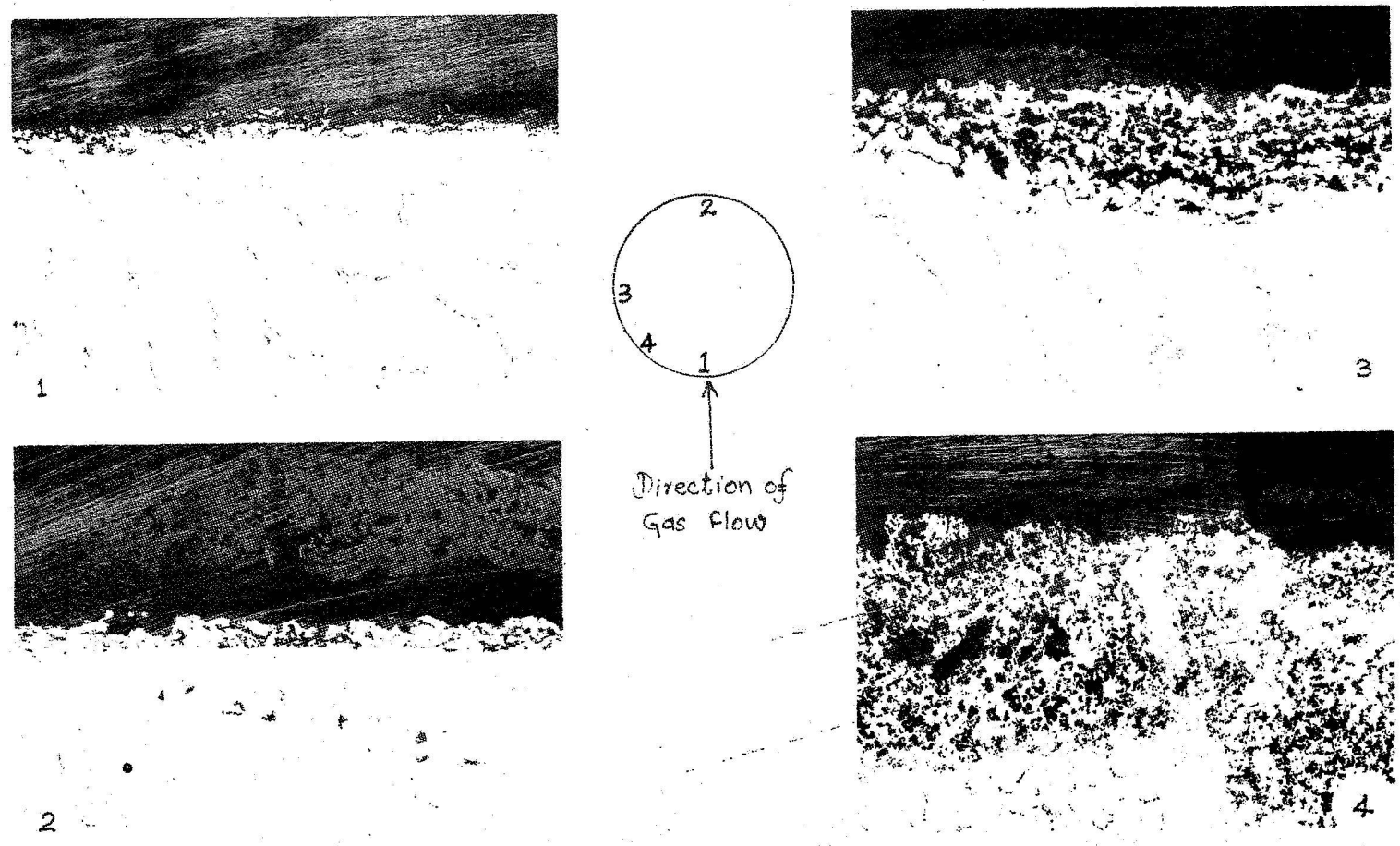


Fig. 16 - Metallographic cross-sections of the NASA-coated specimen after 46.5 hours of exposure; location B (100X).

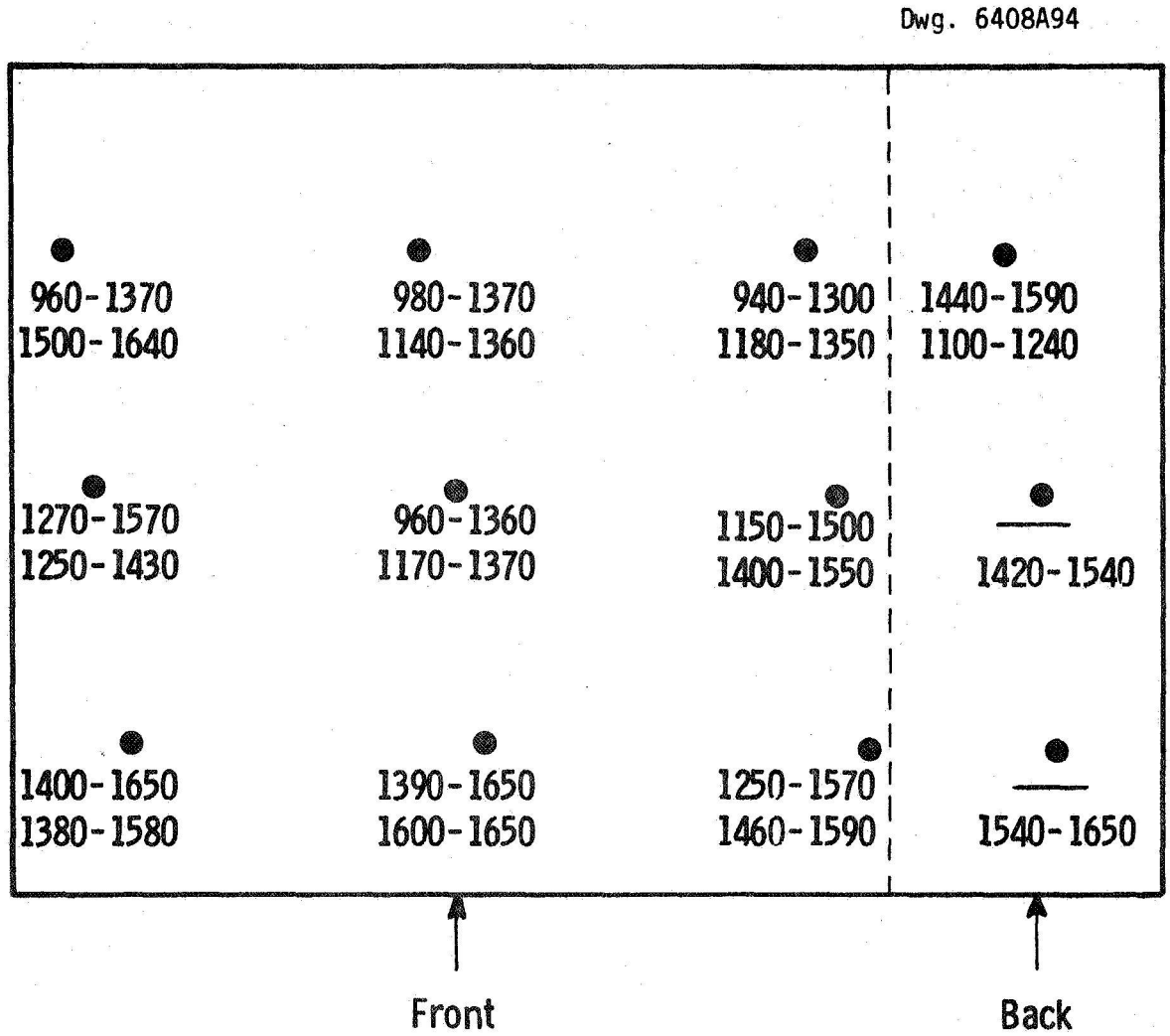
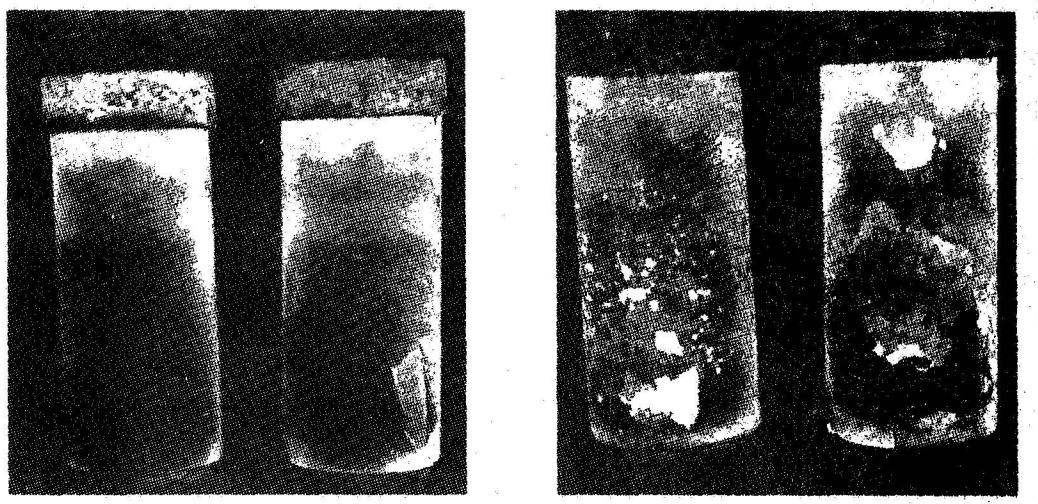


Fig. 17 — Temperatures (in °F) at the metal surface of the air-cooled specimens in the pressurized turbine test passage at 2000°F gas temperature:

Top Numbers - Linde-coated

Bottom Numbers - NASA-coated



NASA LINDE NASA LINDE

(a) 37 hr exposure (b) 63.5 hr exposure

Coated U-500 air-cooled specimens after exposure in pressurized turbine test passage under following conditions:

Gas Temperature	:	2000°F
Gas velocity	:	500 fps
Max. Metal Temp.	:	1650°F
Avg. Metal Temp.	:	1450°F
Pressure	:	3 atm.
Fuel	:	GT-2 diesel + 5 ppm Na + 2 ppm V + 0.5 wt% S.

Fig. 18 - The NASA- and Linde-coated specimens after exposure for different periods at 1650°F maximum metal temperature.

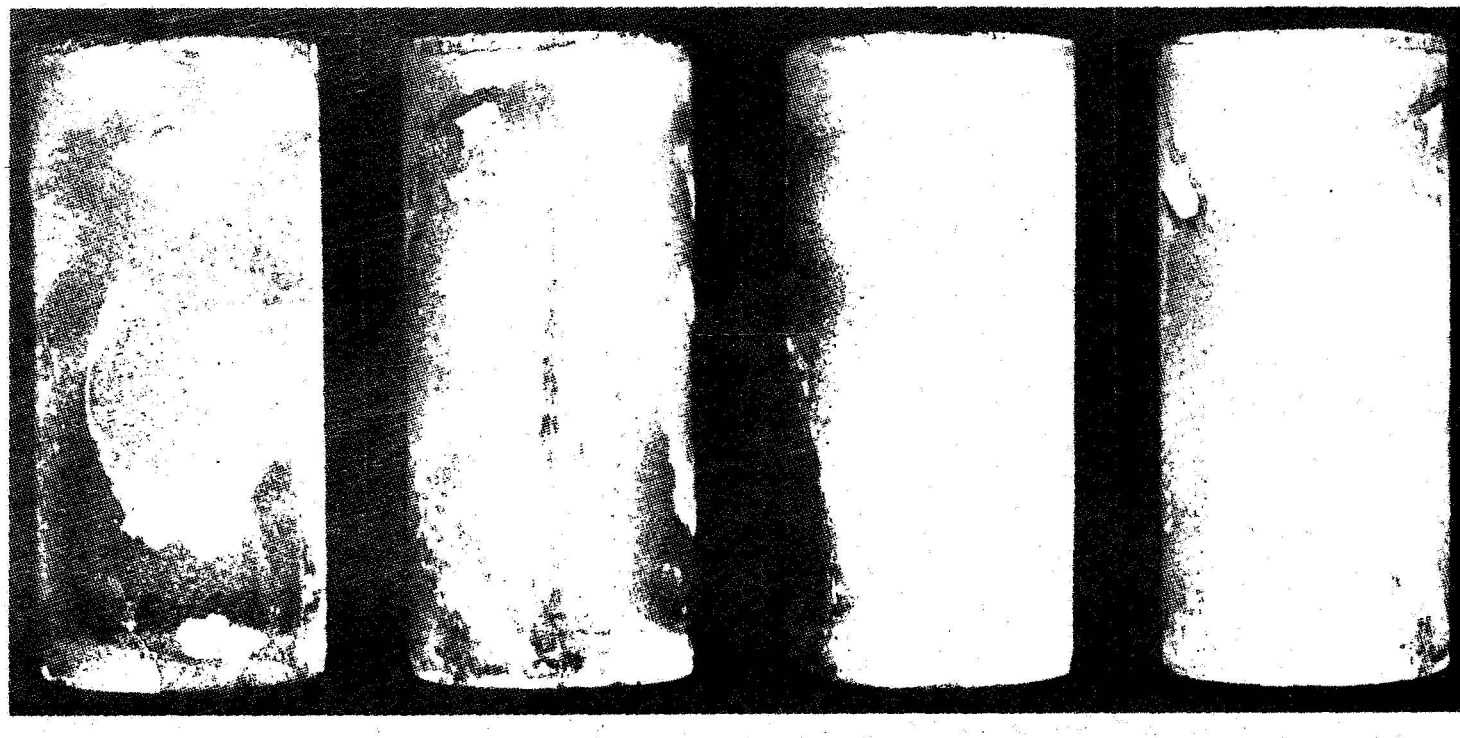


Fig. 19 - Four views of the Linde-coated specimen after 63.5 hours of exposure at 1650°F maximum metal temperature.

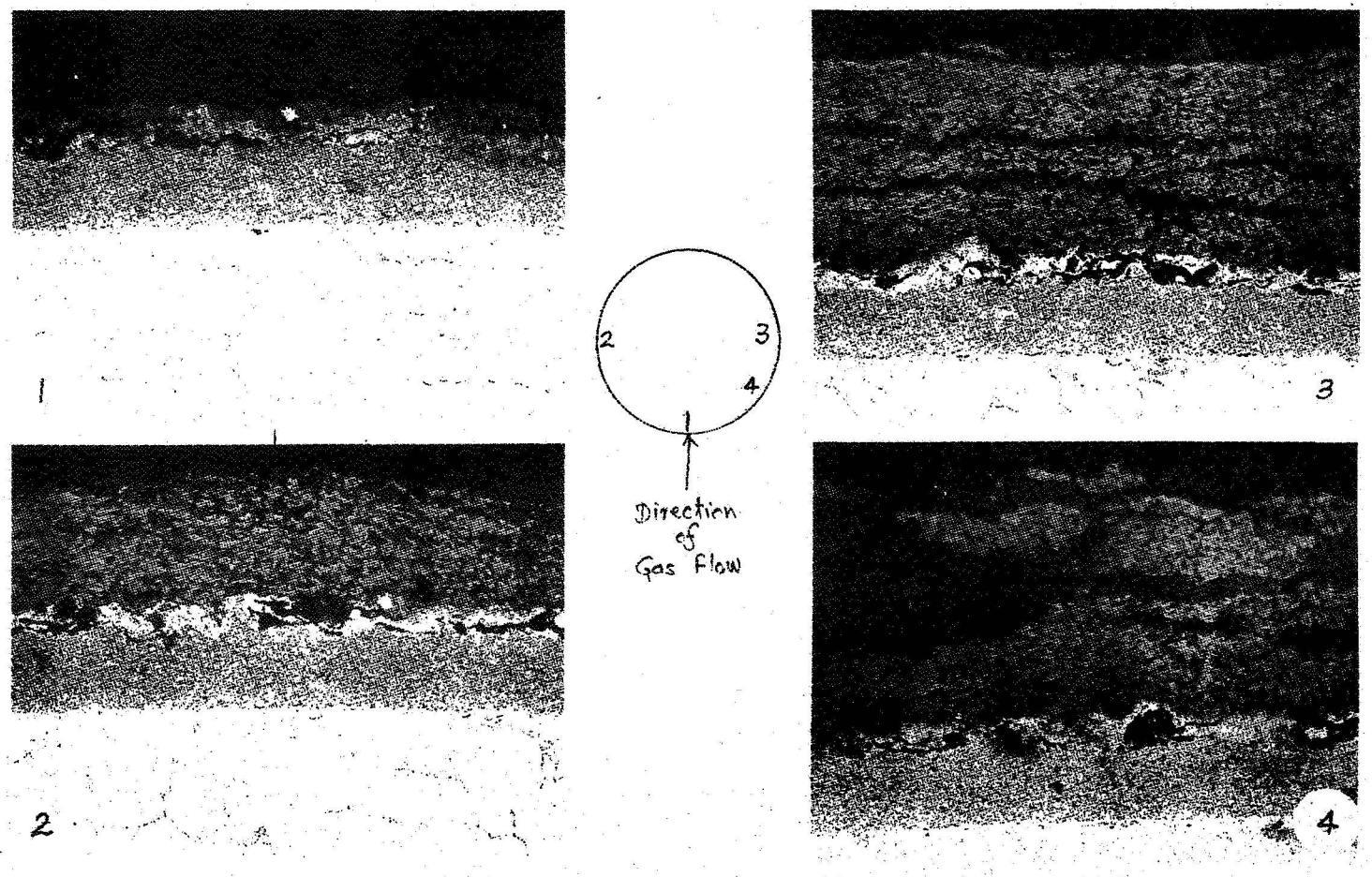


Fig. 20 - Metallographic cross-sections from four different regions of the Linde-coated specimen after 63.5 hours of exposure at 1650°F maximum metal temperature (100X).

Fig. 21 - The NASA-coated specimen after 100 hours of exposure at 1650°F maximum metal temperature.